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APOLLO EXTENSION SYSTEM PAYLOADS

SIMPLIFIED GUIDANCE AND NAVIGATION SYSTEM FOR LUNAR FLYING VEHICLE

Prepared under Contract No. NAS8-20082 by

A. V. Cohee, F. A. Clark, D. F. McAllister, and A. J. Taylor

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For

NASA - GEORGE C. MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

May 1965

ABSTRACT

This report includes a brief analysis of the manual control aspects of a lunar flying vehicle. Environmental factors which affect system selection are discussed. These factors, such as the harsh lighting of the lunar surface coupled with height and distance distortions due to the size of the moon, are analyzed to determine if a manual system represents an improvement over a fully automatic system. Additionally the physiological and psychological considerations are discussed. An important effect is the tilted condition of the vehicle for a hover-translation type of trajectory and how this will degrade the capabilities of the pilot to recognize the target. This tilt angle becomes large, approximately 60°, when the fuel is approaching depletion. A possible solution to this problem is to utilize less than the total number of thrust engines available, particularly during retro when the astronaut may wish to look for a target. A maximum tilt angle of 30 degrees is discussed.

This report concludes that a hover-translation trajectory, when the velocity is optimized, will require less fuel than a ballistic trajectory of the same distance. The equation for the optimum velocity and curves showing the optimum velocity as a function of the distance are shown.

A typical flight is simulated utilizing an IBM computer program in order to determine the reserve fuel remaining at tha target as a function of the mission profiles. Results are shown in graphical form for the fuel used versus coast velocity with altitude as a parameter. It is concluded that for a 1100 ft altitude and a velocity of 900 fps the mission profile will require 497 lbs for a 421 sec flight.

The requirement for maneuvering at the terminal end of the flight is discussed. The report recommends consideration of a simple Rf beacon due to limited time available at the terminus of the flight and due to the large area of surveillance that the pilot must scan.

A system functional analysis is presented for each phase of the mission from liftoff, ascent, and vehicle alignment, to descent. A system description that will meet the requirements is given. It is concluded that the system will weigh 38 pounds, require 62 watts, and occupy a volume of 1990 cu. in.

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For

Systems Concepts Planning Office
Aero-Astrodynamics Laboratory

NASA - GEORGE C. MARSHALL SPACE FLIGHT CENTER

PREFACE

This Technical Report was prepared by the Northrop Corporation for the George C. Marshall Space Flight Center under authorization of Schedule Order No. 4, Appendix F-1 of Contract NAS8-20082. The report represents a consolidation of two previous reports into one comprehensive document containing an analysis of a simplified guidance and navigation system for the Lunar Flying Vehicle element of the Apollo Extension System Payload. The technical work covered by the prior documents was preformed under authorization of Task Orders N-54 and N-63 of Contract NAS8-11096. The resulting technical reports, titled "Apollo Extension System Payloads Analysis of Lunar Flying Vehicle Guidance and Navigation System", NSL E30-33 and E30-53, are consolidated herein.

The NASA Technical Representative for this effort was Mr. Lynn Bradford of the MSFC Aero-Astrodynamics Laboratory.

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SECTION 1.0

INTRODUCTION

The prime objective of the present Apollo program is to place two men on the lunar surface and to return them safely within this decade. Logically, this achievement is a beginning rather than an end, as more extensive operations will follow.

For more extensive lunar operations, a number of "Apollo Extension System" (AES) missions are being considered which will require two flights, one for the delivery of equipment and the other for the delivery of men (Apollo).

Early lunar surface missions of the AES program may be based upon modifications of the LEM to provide a LEM-Shelter (SHELAB) and a LEM-Taxi which will land two astronaut-scientists to occupy the SHELAB for exploration periods up to two weeks. Foreseeable lunar surface scientific missions conducted from the SHELAB require the addition of local mobility for adequate coverage and traverses to lunar surface features removed from the landing site.

One such scientific mission would require an AES payload consisting of a MOLAB, Scientific Equipment, and a Lunar Flying Vehicle (LFV). This payload would be carried to the lunar surface by a LEM Descent Stage.

The lunar flying vehicle is intended to have several purposes. These include: (1) service as an emergency vehicle for the two man crew to transport themselves from a disabled or otherwise unoperative MOLAB back to the LEM, (2) as a transportation vehicle to carry men or materials on planned scientific missions, and (3) as a vehicle to be used in reconnaissance and on observation trips.

The report to follow constitutes a preliminary study of a guidance and navigation system for a lunar flying vehicle.

The lunar flying vehicle under consideration in this study is considered to have the range capability up to 50 miles and to perform according to the following guidelines:

These guidelines, as modified, are as follows:

(1) Known point of origin and termination, error in knowledge of position not included in error analysis.

(2) Gross vehicle weight 1,600 lbs. (including astronauts).

(3) Weight of guidance and navigation system less than 50 lbs.

(4) Propulsion five engines of 100 lbs. thrust each throttleable but not gimballed. Up to three of these engines may be shutdown while retaining full control of the remaining engines.

(5) Vehicle is supplied with a stabilization system which will accept external positioning commands.

(6) Operate lunar day or night

(7) Flight to be handled by one astronaut.

(8) Total ΔV available 4,700 ft/sec.

(9) Maximum operating range 50 statute miles.

(10) Total transit time less than 15 minutes.

(11) Landing point 0.25 mile from target (2).

(12) Target visibility 2.5 miles (day or night).

(13) Manual vehicle control desired.

It has not been possible to follow all of these guidelines completely, and it was necessary to make certain additional assumptions to aid in the mechanization. These are reported as required in the text.

SECTION 2.0
MISSION ANALYSIS

In its application, the lunar flying vehicle will be the lunar based counter-part of a small two-place aircraft. The analogy cannot be carried too far but this is enough to suggest that a purely manual guidance, navigation, and control system may be used. Experience with aircraft control and navigation has proven that a manual system is best, particularly at critical times such as takeoff and landing. As the environment becomes more severe (rain, fog, etc.) the peripheral equipment required, to aid the pilot in making his decisions, increases. In view of the alien lunar environment, careful study is required to determine that a manual system with all required displays and sensors is truly smaller, lighter, and more reliable than an automatic system. With the high cost of delivery, per pound, to the lunar surface and the high price to be paid for failures, it is imperative that every known factor be considered in system design as well as in hardware design.

Factors which must be considered in the system design are:

(a) Environmental effects

(b) Physiological and psychological considerations

(c) Trajectories

(d) Accuracy requirements

Each of these factors will be discussed in detail below.

2.1 <u>ENVIRONMENTAL EFFECTS</u>

Design of the navigation and control system is influenced by the following lunar environmental conditions.

(a) Temperature range of operation ($\approx$ 500° F).

(b) Outer space vacuum ($\approx 10^{-16}$mm Hg pressure).

(c) Harsh lighting conditions during daylight hours.

(d) Condition of the lunar surface.

(e) Small physical size of the moon.

These environmental effects may influence the system design directly, such as temperature control problems; or indirectly through psychological or physical influences upon the astronauts which reflect back into the system design.

2.1.1 Thermal Effects

Thermal effects primarily influence the hardware design and packaging. The high daytime temperatures dictate the use of a minimum of heat producing elements and a restriction upon the power dissipation. The low night temperatures restrict the use of elements requiring close temperature control. In addition, power is not available from the sun during the night, and the astronauts power requirements for heating purposes will be high during this time. The conclusion is that the overall power requirements of the guidance and navigation system must be held to a minimum for both day and night operation.

2.1.2 Lighting Conditions

It will require considerable experience to correctly interpret the visual picture of the lunar landscape under the harsh lighting conditions prevailing in daylight hours. (See References 1 and 2). The total black of shadows will probably be intensified by the filters required by the otherwise high surface illumination. A mountain on the sunward side of the astronaut during morning or evening hours may be visible only in profile against the stars. Viewed from the opposite direction, the mountain will tend to disappear due to flat lighting. The Ranger photographs, even those taken at short range, show curious optical illusions in which craters become domes and rilles become ridges.

To be safe, the navigation system should not rely upon lunar terrain features for azimuth orientation. Celestial bodies such as the sun, Polaris and Canopus may be considered stationary for the LFV missions and make excellent azimuth references.

It may at times be necessary to take off in the vehicle from a position in the shadow of a mountain or other surface feature. Study of the lighting conditions existing in the shadow indicates that it would be difficult to determine without reconnaissance the vehicle position relative to the feature creating the shadow. For this reason, the ascent trajectory should be vertical to avoid possible impact.

2.1.3 Lunar Surface Conditions

The Ranger photographs of the moon show smoothly rounded surfaces. This is true even in the highest resolution pictures from Ranger IX with a resolution of 10 inches. This strongly suggests substantial dust layers or a loosely consolidated surface. Present design thinking must accept the possibility that when the lunar flying vehicle takes off it may be surrounded by a thick dust cloud up to a considerable altitude ($<$ 500 ft?). For this reason, azimuth information must be stored within the vehicle at least until it has climbed above the dust cloud. During this vertical ascent, the vehicle should be rotated in yaw to align the roll axis with the desired azimuth. This rotation should be initiated immediately after take off to give the pilot time to check azimuth orientation optically prior to the start of horizontal motion.

2.1.4 Physical Size of the Moon

The moon has a diameter less than one fourth that of the earth. As a result the horizon will be nearer, on level ground, and more important, the visual scale of distances and heights will be distorted. (Reference 1). This will provide a psychological barrier to full manual control of the vehicle in mountainous areas. Until adequate maps containing accurate vertical profiles are available, the only safe trajectories will be at altitudes providing substantial terrain clearance. This reduces the available fuel reserves for terminal maneuvers but is probably less costly than a climbing maneuver at high velocity.

2.1.5 Indirect Effects of the Lunar Environment

The physical effects on the astronauts of direct exposure to the lunar environment need not be discussed here. It is sufficient to say that the guidance and navigation system must be designed for the highest possible reliability in performance of its required functions. Psychological side effects such as were discussed under 2.1.2 and 2.1.4 may be sufficient to preclude full manual control of the LFV, at least until the astronaut has gained experience in the visual interpretation of the lunar landscape.

2.2 PHYSIOLOGICAL AND PSYCHOLOGICAL CONDITION OF THE ASTRONAUTS

Various psychological effects which influence the system design have been mentioned in the preceding paragraphs. In this section one concern is for the condition of the astronauts in an emergency return from the MOLAB to the LEM vehicle.

An MOLAB mission would be aborted in this manner only if the MOLAB were partially or totally disabled or if one or both astronauts were themselves partially or totally disabled. Even though it is the MOLAB vehicle which is incapable of return, the circumstances surrounding the failure may result in partial disability of the astronauts. For example, an electrical fire aboard the MOLAB may release toxic fumes which reach the astronauts air supply.

Whatever the cause, there is a substantial probability that in an emergency the astronauts will be capable of only minimal tasks. A safe return to the LEM under these conditions will require at least a semi-automatic mode of operation of the guidance and navigation system.

Another psychological effect which must be considered in the guidance system design is engendered by the vehicle design. The thrust is applied along the vehicle yaw or vertical axis. To give the vehicle a horizontal velocity, it is necessary to tilt the yaw axis from the vertical. If a constant altitude is desired during the horizontal thrust period, the vertical component of thrust must just equal the lunar gravitational attraction. For the high accelerations required for efficient fuel use, the tilt angle becomes large ($> 60^{\circ}$), particularly when the fuel is approaching depletion.

These large tilt angles can be quite annoying, particularly since the vehicle structure has been reduced to an absolute minimum for weight reduction, and the astronauts will be literally hanging by their seat belts when in this position. To add to the problem, the astronauts will want to be looking for their target during the later phases of the retrofire when the tilt angle is most pronounced. It is therefore desirable to restrict the tilt, especially during retrofire, by throttling back on the engines at the expense of fuel efficiency.

2.3 TRAJECTORIES

Basic design features of the guidance, navigation and control system are dictated by the use of rocket propulsion. Selection of the mission profile is controlled to a major degree by the high rate of fuel consumption and the high acceleration attained. These two factors together dictate the use of a high translation velocity for the most efficient use of fuel. With high velocities, changes in course become expensive in fuel. The guidance and navigation problem therefore resolves itself to an orientation of the vehicle in azimuth so as to arrive at the destination by a straight line maneuver using a programmed thrust profile such that there will be sufficient fuel remaining to correct for azimuth and initial position errors at the end of flight. The absolute requirement for the most efficient use of fuel is only valid for maximum range during an emergency mission. The system which will control this mission, however, will safely control any mission required so long as it is broken down into straight line segments.

The question which will be explored here is the selection of the trajectory which provides the most efficient use of fuel and at the same time satisfy the constraints dictated by the environment factors already discussed.

Two possible modes of flight present themselves. The first is a ballistic mode. In this mode thrust would be maintained at some constant angle from the vertical until the desired velocity was reached, then the engines would be shut down during a free-fall period after which they would be re-ignited for a retro-thrust phase. This gives a trajectory of the same form as used in an ICBM.

The second method of transport is the hover-translation mode. In this method, the vehicle is boosted vertically to an altitude which will clear intervening terrain. It is then tilted over at an angle and full thrust is applied until the vehicle reaches the desired horizontal cruise velocity. The angle of tilt is chosen so the vertical component of thrust just nullifies the lunar gravitational attraction. When the cruise velocity is achieved, the vehicle is erected to the vertical and the engines throttled back to nullify gravity again. Near the terminus, a retro-fire phase is used to zero the horizontal velocity, followed by vertical descent to the landing site.

The ballistic mode of transport was initially found to be attractive due to lower fuel requirements. The studies have shown, however, that an optimum velocity exists for hover-translation, and that flights at this velocity require less fuel than the ballistic trajectory for the same distance. While constant angle thrust can be used for the ballistic case, velocities must be controlled quite accurately to minimize down-range miss. Accelerometer scale factor errors contribute twice as much error in ballistic flight as in hover flight. Vertical velocity errors contribute to miss by changing the ballistic time of flight while at worst they waste fuel in hover-translation, but do not contribute to miss. Because ballistic flight is relatively intolerant of errors, manual control would be ruled out, and sufficient computer must be supplied to solve the complete problem automatically. Another fault to be found with ballistic flight is that the engines are shut down, then re-ignited. If, for some reason, re-ignition did not occur or full thrust were not achieved, the vehicle would already havefrom 500 to 600 fps downward velocity making escape nearly impossible. The fuel consumption for a 50 mile flight in the ballistic mode is shown in the last column of Table 2.1.

It can be shown (see Appendix B) that the optimum velocity for the hover-translation cruise is given by

$$V_{opt} = \sqrt{\frac{g_m S(1-L_a)(1-L_d)}{2-L_a-L_d}} \tag{1}$$

where V_{opt} = optimum velocity

g_m = moon gravity

S = distance covered

L_a = gravity and turn losses during ascent

L_d = gravity and turn losses during descent

The results of evaluating the expression for $L_a = L_d = 0$, $L_a = L_d = 0.1$, and $L_a = L_d = 0.2$ are given in Figure 2.1 for ranges up to 50 statute miles.

A convenient measure of the relative efficiency of various mission profiles is the reserve fuel remaining at the target. In order to estimate this reserve fuel, a program has been written for the IBM 1620 computer to simulate typical flight. The equations used for the simulation are given in Appendix A. The results are quite conservative because non-optimum thrust programs are used for ease of programming. The LFV is assumed to rise vertically to the desired altitude. This phase assumes maximum thrust for a period of time, then no thrust while the vehicle coasts up to the point of zero vertical velocity. At this time the angle of thrust is computed which gives a vertical component of acceleration equal to lunar gravity. This angle is assumed constant while thrust is continued until the desired velocity is reached. In the actual case, the thrust angle, measured from the vertical, would increase slowly if constant thrust were used during this phase. The reason, of course, is that as fuel is used up the acceleration increases when the thrust is held constant. This results in the use of less fuel to achieve the specified velocity. Transition from thrust to horizontal coast is assumed to be instantaneous. During coast, the engines are throttled down to null the moon's gravity. This requires the shut-down of one engine since the minimum thrust of five engines is too large. This fifth engine does not have to be reignited during the remainder of the flight. Since the retro-fire phase requires excessive angles at full thrust, a flow rate of 1.0 lb/sec is used for retro. Again, the angle necessary to null lunar gravity is assumed to be held during the entire retro-fire phase.

Descent is computed by assuming the same thrust is used as in retro-fire preceded by a period of free fall. The thrust is applied long enough to null the velocity gained in falling from the initial altitude. In an actual flight, the engines would be throttled back to a minimum flow rate, with possible shut down of two additional engines. This would permit the vehicle to accelerate

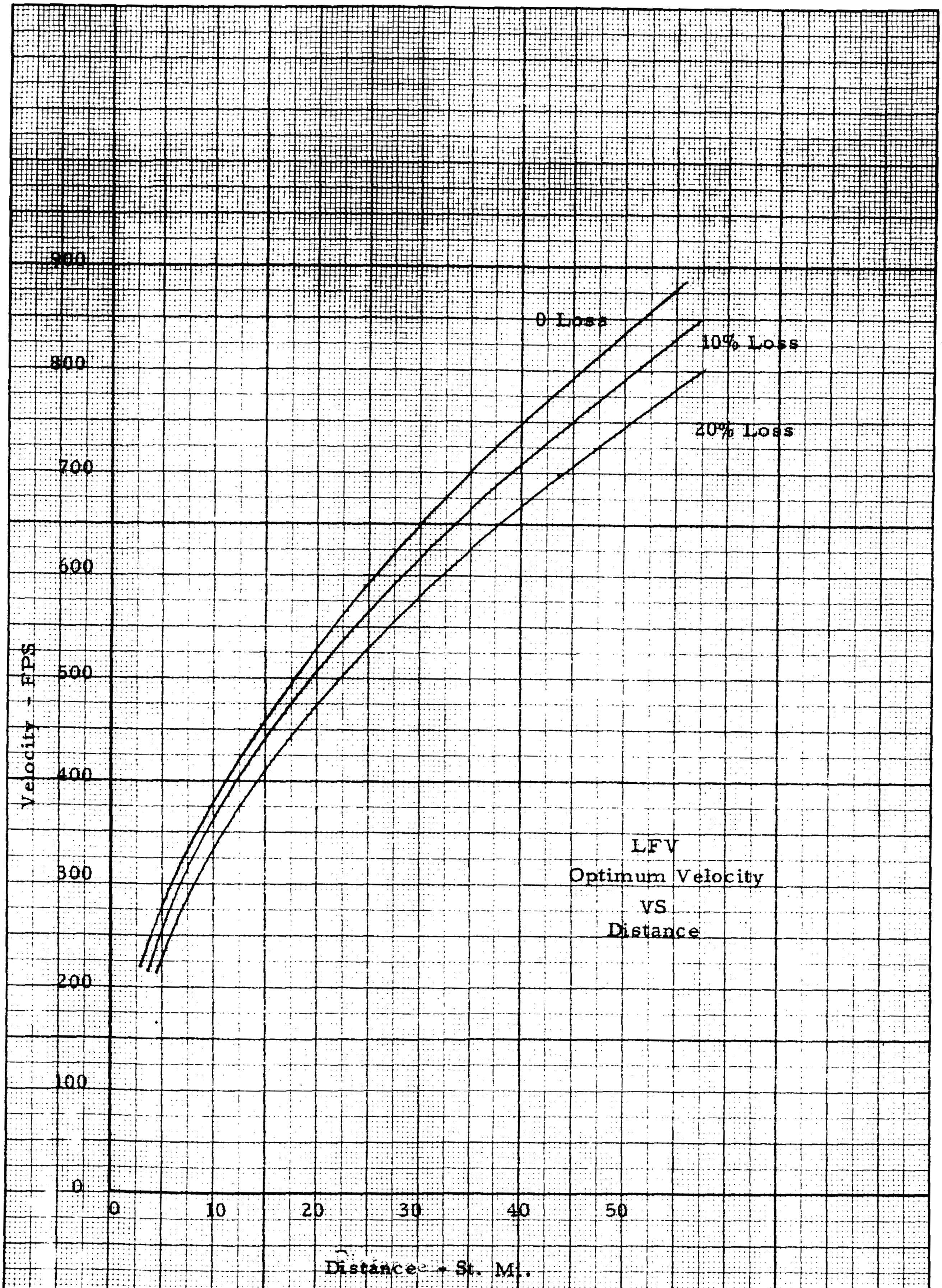

900
800
700
600
500
400
300
200
100
0
0
10
20
30
40
50
0 Loss
10% Loss
20% Loss
Velocity - FPS
LFV
Optimum Velocity
VS
Distance
Distance - St. Mi.

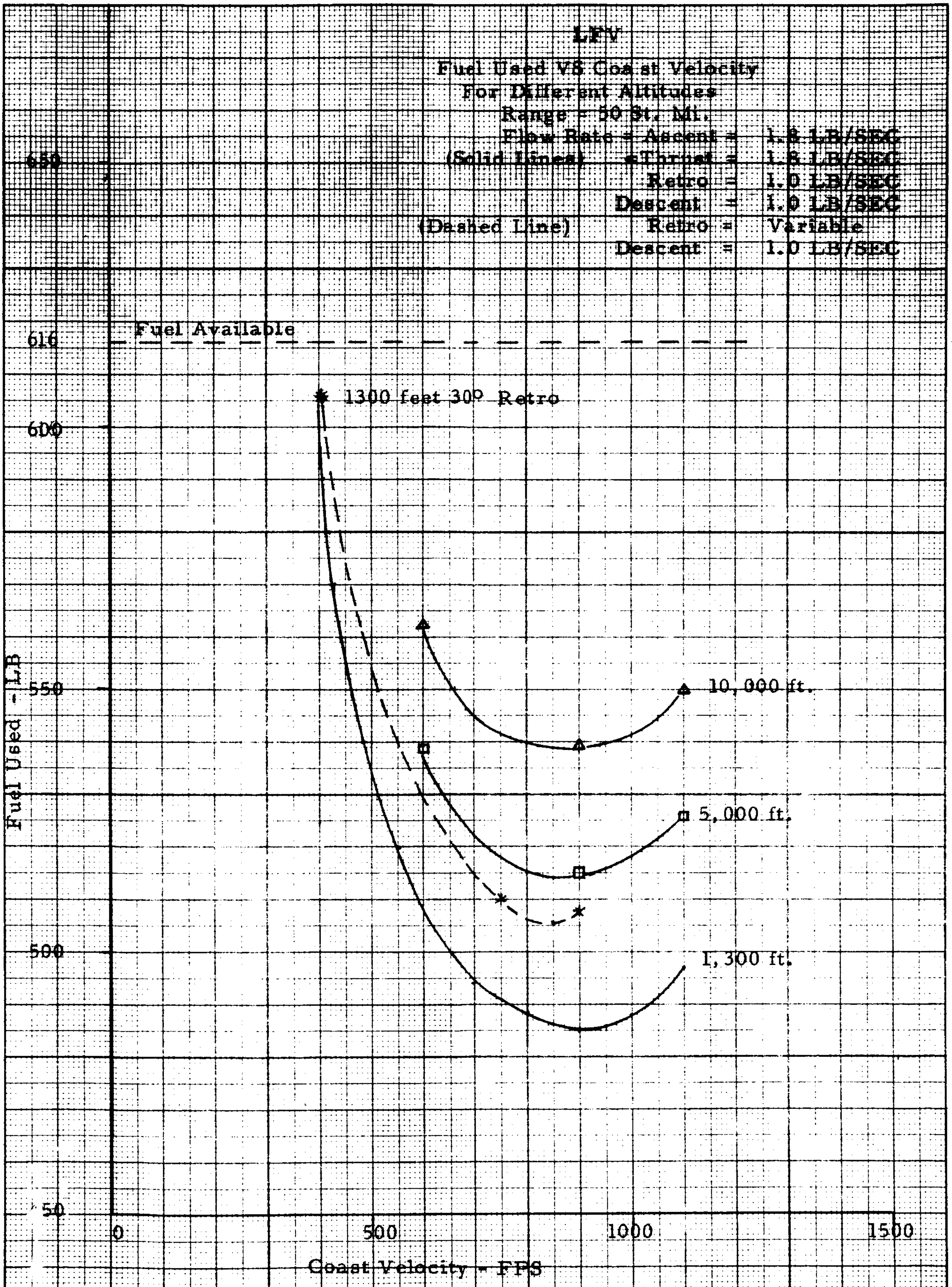
Figure 2.1

downward to a suitable velocity of fall without requiring engine reignition. The fuel required for traveling 50 miles at different altitudes and coast velocitites is given in Figure 2.2. A breakdown of fuel consumption is also given in Table 2.1. Since thrust angles near 45° are required when the flow rate is 1.0 lb/sec during retro-thrust, some trajectories were simulated which reduced the angle to 30°. The resulting flow rate varied from 0.7 lb/sec at 400 fps to 0.85 lb/sec at 900 fps. The dashed line in Figure 2.3 shows the fuel consumed at 1300 ft. altitude for this 30° tilt angle. The increase in fuel consumed is 24 pounds for 1300 foot altitude or 27 pounds for 10,000 foot altitude.

The computational results show that the thrust angle (vehicle tilt angle) is quite large. During the initial boost, this tilt angle is 61° from the vertical for full thrust. If full thrust were used during the retro phase, the tilt would be even greater. While the tilt angle may not be a problem during boost, it is definitely a problem during retro when the astronaut may wish to look for the target. The increase in fuel consumption resulting from constraining the angle of retro to 30° is probably justified.

Maneuvering at the terminus is considered to be a necessity. Even if the 2-sigma accuracy of 0.25 miles can be met, a reserve sufficient for a one mile flight should be available at the end of the descent phase, for this 0.25 mile error does not include initial position error. A reserve sufficient for 5 miles of maneuvering would be much more comfortable. Figure 2.3 shows the distance which can be covered in a hover-translation type of maneuver with a given amount of fuel. For comparison, the theoretical maximum distance assuming no energy loss due to gravity during the thrusting phases, and an empirically determined distance using a flow rate of 0.7 lb/sec are shown. The empirical curve includes gravity losses.

Hand computation shows that certain modifications of the hover-translation mode may prove to be more efficient in fuel consumption. For one thing, a transition phase between ascent and boost using a constant pitch rate may save up to 20% of the boost fuel required. In addition, a constant pitch-over during the translation resulting in a continuous acceleration during the entire cruise period gives an additional possible savings. Proof of the increased efficiency will require a much more sophisticated computer program requiring the IBM 7090 computer for its solution. After this more efficient mission profile is defined, it will still be necessary to examine the mechanization requirements to determine whether these can be satisfied without undue complications.



LFV
Fuel Used VS Coast Velocity
For Different Altitudes
Range = 50 St. Mi.
Flow Rate = Ascent = 1.8 LB/SEC
(Solid Lines) Thrust = 1.8 LB/SEC
Retro = 1.0 LB/SEC
Descent = 1.0 LB/SEC
(Dashed Line) Retro = Variable
Descent = 1.0 LB/SEC
Fuel Available
1300 feet 30° Retro
10,000 ft.
5,000 ft.
1,300 ft.
Fuel Used - LB
650
616
600
550
500
50
0
500
1000
1500
Coast Velocity - FPS

Figure 2.2

TABLE 2.1

	1300 ft Altitude			5000 ft Altitude			10,000 ft Altitude			Ballistic
Coast Velocity, (FPS)	600	900	1100	600	900	1100	600	900	1100	828
RISE										
Time, sec	15.0	15.0	15.0	29.10	29.10	29.10	40.80	40.80	40.80	15.00
Fuel, lb	27.00	27.00	27.00	52.38	52.38	52.38	73.44	73.44	73.44	27.00
THRUST										
Time, sec	59.87	88.25	106.62	58.63	86.43	104.42	57.61	84.94	102.63	118.00
Fuel, lb	107.77	158.85	191.91	105.53	155.57	187.96	103.71	152.89	184.73	212.40
Distance, S. Mi	3.36	7.39	10.87	3.29	7.24	10.64	3.24	7.11	10.46	9.12
Angle deg fm vert.	61.19	61.19	61.19	61.69	61.69	61.69	62.11	62.11	62.11	37.00
COAST										
Time, sec	362.00	168.00	85.00	364.00	172.00	89.00	365.00	175.00	92.50	206.95
Fuel, lb	265.13	125.11	63.23	262.24	125.95	65.11	259.39	126.35	66.72	0.0
Distance, S. Mi	41.14	28.64	17.71	41.36	29.32	18.54	41.48	29.83	19.27	
RETRO										
Time, sec	95.52	160.09	201.52	92.76	154.66	194.48	90.60	150.41	188.89	103.06
Fuel, lb	95.52	160.09	201.52	92.76	154.66	194.48	90.60	150.41	188.89	185.5
Distance, S. Mi	5.50	13.95	21.57	5.34	13.47	20.81	5.22	13.09	20.20	7.94
Angle, deg fm Vert	-48.56	-44.70	-43.39	-49.41	-45.72	-44.45	-50.09	-46.54	-45.32	37.00
Flow Rate, lb/sec	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.8
DESCENT										
Time, sec	13.38	13.67	13.52	25.66	26.24	25.97	35.65	36.45	36.10	
Fuel, lb	13.38	13.67	13.52	25.66	26.24	25.97	35.65	36.45	36.10	99.0
TOTAL										
Time, sec	545.78	445.02	421.67	570.16	468.43	442.98	589.67	487.60	460.92	443.01
Fuel, lb	508.81	484.74	497.20	538.59	514.81	525.92	562.80	539.56	549.89	523.9
Fuel Reserve, lb.	107.19	131.26	118.80	77.41	101.19	90.08	53.20	76.44	66.11	92.1

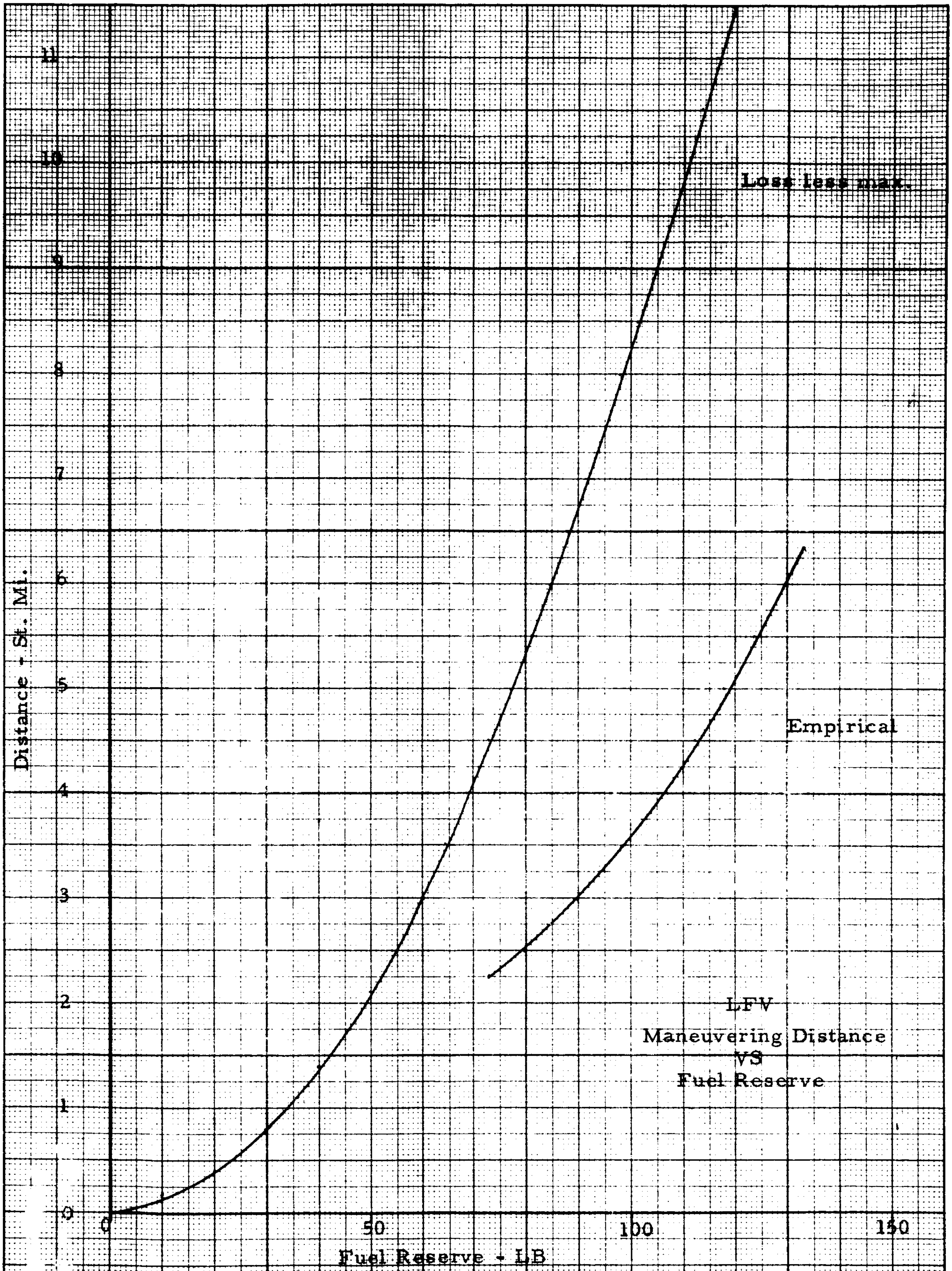

Loss less max.
Empirical
LFV
Maneuvering Distance
VS
Fuel Reserve
Distance - St. Mi.
0
1
2
3
4
5
6
7
8
9
10
11
0
50
100
150
Fuel Reserve - LB

Figure 2.3

2.4 ACCURACY REQUIREMENTS

The relationship between mission accuracy requirements has been derived in Appendix C. This relationship will be used in future LFV work and is included herein for reference purposes.

It is not at all certain at this juncture that the stated accuracy requirement of 0.25 miles (2 sigma) can be met. The requirement is not difficult to achieve under earth conditions. When one considers the environmental conditions under which both the equipment and the astronauts are working, it is easy to see that even the best design can fail due to unknown factors. When the unknowns in initial position determination are added and the overall error is considered, it can be appreicated that the total 2.5 mile visibility range of the LEM may be required. This means that at the end of the horizontal translation, the astronauts may have a circle of 2.5 miles radius to search for the LEM, or a total search area of more than 18 square miles.

Under night time conditions an optical beacon on the LEM would make the search quite simple. Search under daytime conditions however may be quite another story. Previous studies at Northrop have indicated that the harsh lighting during daytime hours make for difficult seeing conditions. (Reference 2). The filters required when the sun is in the range of vision will provide particular problems, and it is even possible that a shiny area of surface may provide a false location.

When one further considers the time available for the search, the problem becomes even more critical. Assuming a travel distance of 50 miles at an altitude of 1,300 feet, the vehicle has fuel sufficient to hover stationary for approximately 3 minutes before all fuel is expended. In the worst case then, the astronauts must visually search 18 square miles of terrain and leave fuel sufficient for a possible 2.5 mile flight, this gives only 90 seconds to scan the area. During the later portion of the retro phase, the vehicle is within visual range and this period may be used for search. Even so, the search time is extremely short.

For these reasons, it is recommended that consideration be given to the inclusion of a simple RF interrogator. This need have only 100 to 200 milliwatts of power, and a simple horn antenna to give a beam width of 60 to 90 degrees would be adequate. The frequency could be the same as used for the LEM rendezvous beacon. This line-of-sight interrogator would greatly increase the speed of location of the LEM and even increase the range of visibility. At the same time the error specification on the guidance and navigation system could be relaxed. In so doing the success of manual operation would be more certain.

SECTION 3.0
SYSTEM FUNCTIONAL ANALYSIS

3.1 COORDINATE SYSTEM

The coordinate system to be used in the following follows aircraft practice. (See Figure 3-1). The yaw axis is a vertical axis with positive pointing down. The vehicle thrust is parallel to the yaw axis and the resulting acceleration is negative. The roll axis is the fore-aft axis with the positive direction forward. The pitch axis is across the beam and is orthogonal to roll and yaw. Thus, the positive pitch axis will be on the right when facing the positive roll direction.

3.2 VEHICLE DYNAMICS

In the absence of vehicle specifications, it is necessary to make some assumptions regarding the vehicle to provide a basis for guidance design.

(a) Vehicle is stabilized by an unspecified system for zero net torque around the C.G.

(b) There are no thrust components parallel to either the roll or the pitch axis.

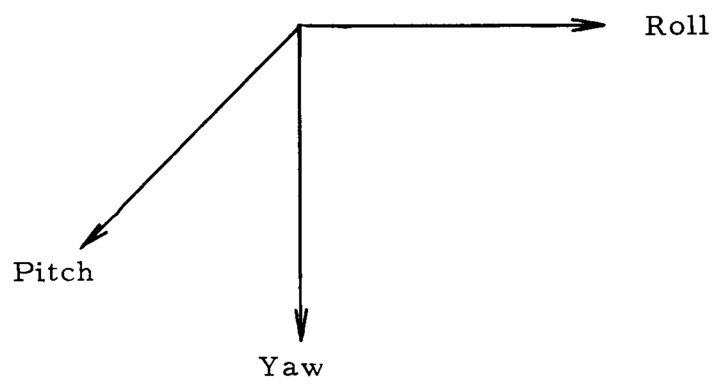
(c) The vehicle control system is a three-axis system providing individual control of vehicle rotation around each axis.

(d) The vehicle control system will accept bipolar on-off commands to torque the vehicle. Torquing proceeds at a constant rate while the command is present.

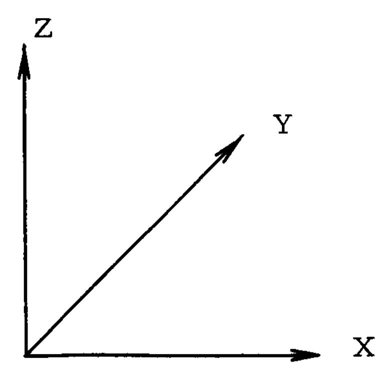
(e) Actuators are available for altitude control even when the engines are cut off.

(f) Thrust level can be controlled by external signals over a fuel flow range from 0,6 lbs/sec to 1.8 lbs/sec. In addition, three of the five engines may be shut down completely while retaining throttle control over the remaining two engines.

(g) The poles and zeros of the stabilized vehicle closed loop transfer function can be compensated by simple networks to provide a stable closed loop guidance system with the desired characteristics.



Control Coordinate System



Trajectory and Error Analysis Coordinate System

Figure 3.1

3.3 OPERATIONAL SEQUENCE

The guidance and navigation system design must accommodate the following mission phases:

(a) Liftoff, ascent and vehicle alignment

(b) Transition to translatory flight

(c) Translation boost

(d) Coast

(e) Retro

(f) Terminal maneuvers

(g) Descent

In addition to the above, the guidance and navigation system must itself be aligned prior to liftoff.

3.4 DETAILED ANALYSIS OF MISSION PHASES

3.4.1 Guidance and Navigation System Alignment

Essentially, alignment consists of determination of the direction of the local vertical and of the azimuth reference. Without a platform the local vertical can be located by the use of a vertical gyro, by a plumb bob, or by astronomical observations. Either of the first two methods will be subject to some error until the exact gravitational figure of the moon is determined. The last method can determine the direction of the selenodetic vertical, and the position on the lunar surface, however, the necessary observations and computations are cumbersome and time consuming. Everything considered it is hardly a method requiring minimal tasks, as required in paragraph 2.2. A composite instrument consisting of a telescope mounted on a tilting base provided with bubbles for leveling in pitch and roll, or mounted on pendulous pitch and roll gimbals, can provide automatic readout of alignment angles in all three axes and will require a minimum of manual manipulation. An additional advantage will be inclusion of redundant manual readout of angles and the possibility of establishing local vertical and position by astronomical observation in the event of failure of the leveling mechanism.

Once the angles required to align the vehicle to the vertical and the azimuth reference are determined they can be read into the computer storage for use in the vehicle alignment phase. During the alignment phase, the target azimuth and range are likewise read into the computer. These inputs are used in preference to position coordinates to ease the computer requirements. The input of target azimuth and range can be an input from the MOLAB computer which is updated periodically. Thus the LFV would be available for emergency use without a wait for computing these quantities.

After the astronauts have become experienced in the vehicle operation it may be possible to dispense with the alignment phase prior to takeoff. The system alignment, vehicle alignment and ascent phase is combined into one. After takeoff, the astronauts manually control the vehicle in pitch and roll until ground sightings show a zero lateral drift rate. The vehicle is then level. The telescope is used to locate the azimuth reference and the vehicle is then rotated to the desired flight azimuth. The operation is thus entirely manual. The method has the disadvantage of inefficient use of fuel if the vehicle hovers during alignment. Substantial errors can be introduced here if the vehicle control system is limit cycling during the alignment. A solution could be a very slow limit cycle or a proportional control system.

3.4.2 Lift-off, Ascent, and Vehicle Alignment

Prior to lift-off, the desired flight altitude is selected. This should be as low as is safe for terrain clearance to give most efficient use of fuel. Several possible ascent thrust profiles exist which require slightly different mechanizations but any of these can be controlled either manually or automatically without undue complication. For example, acceleration can be controlled to a constant value with time being the variable. Alternatively the fuel flow rate can be held constant and accelerometer pulses integrated to the desired vertical velocity. The same hardware is required in either case and the computations are quite simple.

At lift-off the strapdown gyros are placed in inertial control mode. After the desired vertical velocity is attained, the engines are throttled down to minimum fuel flow. At this time, the vehicle is rotated, one axis at a time, to vertical alignment with roll in the desired azimuth direction. (If the vehicle were seriously misaligned from the vertical prior to takeoff the nominal sequence must be altered. Alignment to the vertical would be accomplished during a short hover phase immediately after lift-off).

After the vertical alignment is completed, the azimuth can be checked manually by use of the telescope. The vehicle then continues to coast upward until the altitude for start of the transition phase is reached.

3.4.3 Transition to Translatory Flight

The transition to translatory flight is a controlled pitch-over together with a controlled increase in thrust level. The controlled pitch-over is best performed automatically by torquing the pitch gyro at a constant rate. Since the rate is known, the computer acts as a sequencer and applies the torquing rate for a controlled length of time.

By controlling the pitch rate and the acceleration in this manner, it is possible to precompute the lateral velocity and distance traversed during this maneuver. These two quantities are stored within the computer and are there substracted from the desired values.

An alternate to this transition phase would be to continue the ascent coast until the vertical velocity is reduced to zero by the gravitational acceleration. Then the vehicle is pitched over and the translation boost begun. This is simpler to mechanize but is less efficient in fuel.

3.4.4 Translation Boost

Essentially this consists of the application of thrust at a pitch-over angle such that the lunar gravitational attraction is just nullified. The horizontal component of thrust accelerates the LFV until the desired cruise velocity is attained. Several different thrust profiles can be utilized, and each requires a slightly different mechanization. The efficiency here is very important for the major position of the flight fuel is used in the boost, coast, and retro phases.

The mechanization for this phase requires an accelerometer for control of thrust level and the computer for sequencing and summing.

3.4.5 Coast Phase

During this phase, the vehicle is erected to the vertical and the engines throttled back to nullify lunar gravity. An altimeter is used to provide warning if the terrain clearance dips below the danger point, but it is not suitable for controlling altitude except when flying over a level surface. The thrust axis accelerometer will be used to adjust the thrust acceleration to the required value. If the thrust axis is not exactly vertical, then there will be a vertical acceleration since gravity is not nulled. The downward velocity resulting from a tilt angle of one degree will amount to only 0.2 ft/sec after 200 seconds and this can be neglected.

The more serious error resulting from a nonvertical thrust axis is the lateral position error at the end of flight, (assuming the error is in roll). A one degree roll error continuing during an entire coast phase of 200 seconds gives a transverse error of 1800 feet. If a roll position error of this magnitude exists from the start of boost, the total transverse error at the target from this source alone will amount to more than 4000 feet at 50 miles range. This error is analogous to drift error due to crosswinds in aircraft. It can be measured and corrected using a precision inertial platform but not with the simple mechanization proposed. It may be possible to use a form of drift meter, manually operated, with sightings on the lunar surface to measure and correct for the vehicle tilt, but this measurement is also subject to error due to motions of the vehicle.

The coast phase can either be timed, if the velocity is accurately known, or the measured velocity can be integrated and compared against a preset distance to be traversed. This computation and sequencing is best performed by the computer. At the preset distance the astronauts are signalled to start retro. Automatic sequencing of the retro presents no difficulties.

3.4.6 Retro Phase

The retro phase is quite similar to the boost phase except that one or more engines may shut down to decrease the pitchover angle. Once these engines are shut down they will no longer be required during the flight. The mechanization will be identical to that used for boost.

3.4.7 Terminal Maneuvers

The terminal maneuver may start immediately after the retro phase or may be phase or may be preceded by a partial descent. The first and most important task is to locate the target, and this must be done manually. After the target is located, a second boost, coast, and retro sequence is used to bring the LFV as near as desired to the target. This sequence can be entirely manual aided by the computer.

3.4.8 Descent Phase

The descent phase is virtually identical in mechanization requirements to the ascent. The primary difference is that one or more engines will be shut down completely, for the reason that when the fuel is nearly depleted the minimum thrust of five engines will accelerate the vehicle upward and thus it cannot land. After a 50 mile journey only two engines will be operating during the descent. An altimeter is required for monitoring the altitude and altitude rate during the descent. With this assistance, control can be entirely manual.

SECTION 4.0

SYSTEM DESCRIPTION

One possible mechanization which will meet the requirements of the preceding sections is shown in Figure 4.1. The active elements are described below.

(a) Three single-axis miniature integrating gyros (MIG type) are hard mounted near the cg with the gyro axes aligned with the vehicle axes. The gyros are useable in both rate and position mode. Constant current torquing is used when axis rotations are required.

(b) One single axis accelerometer with digital output. This accelerometer is aligned with the thrust axis.

(c) Computer-sequencer. This is a simple unit with a core memory. Only integration, addition and comparison capabilities are required. A built in clock is used for sequence timing.

(d) Telescope and mount. The telescope is supported in azimuth-elevation gimbals on a small table which is either provided with leveling screws or is supported in a pendulous mounting. The mount is capable of being locked after leveling while the azimuth reference is located. The leveling system is provided with both electrical and manual readout of vertical alignment angles. The telescope azimuth gimbal is likewise provided with an electrical readout.

(e) Altimeter. The altitude range required is from 20 feet to 20,000 feet. Manual readout of altitude and altitude rate is provided. An alarm is provided if the altitude drops below a preset value during boost, cruise, and retro, or if the altitude rate is too great.

(f) Displays and Manual Controls. These include complete manual controls for vehicle operation, display of all important parameters, and a keyboard for communication with the computer.

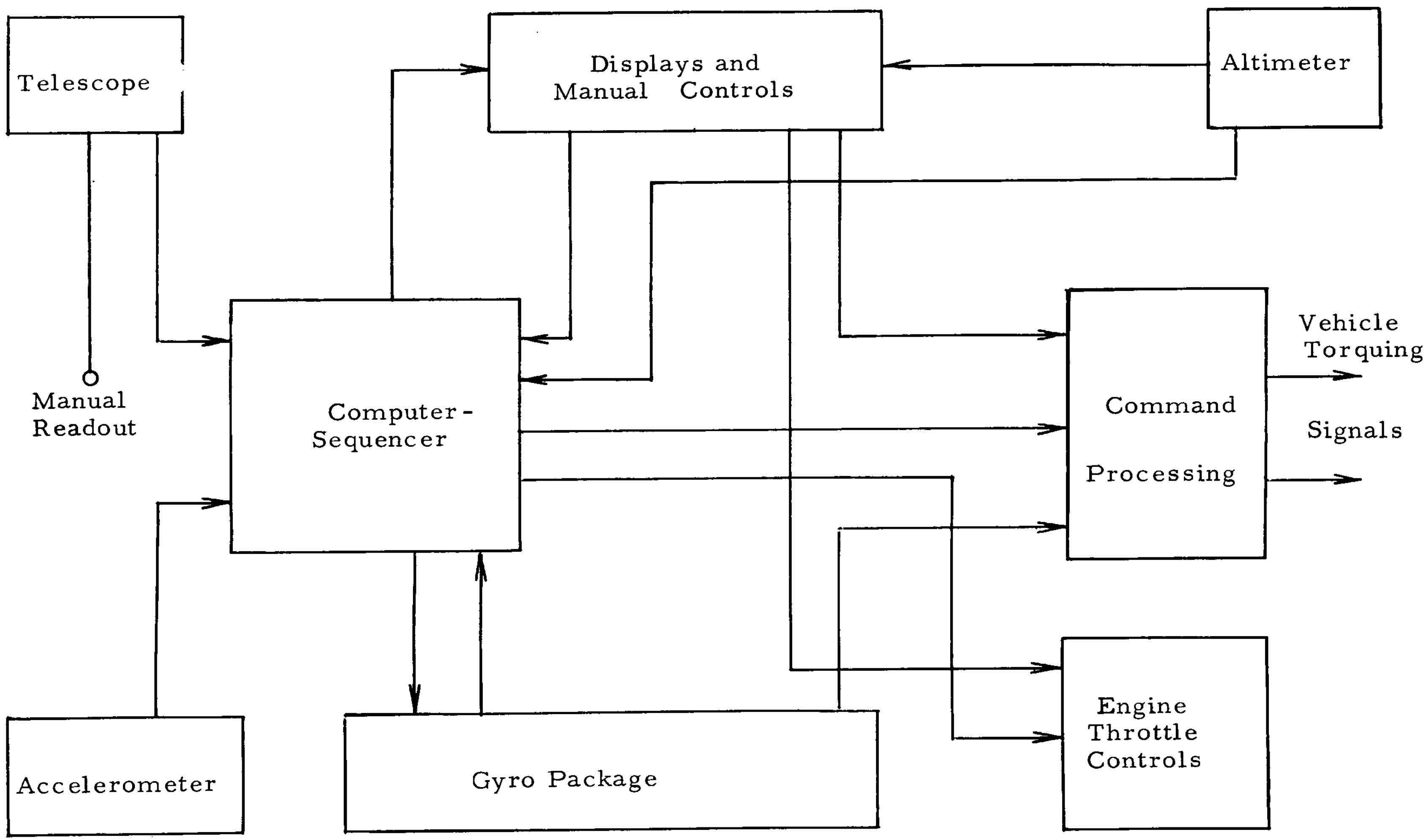


FIGURE 4.1 System Functional Block Diagram

4.1 SYSTEM OPERATION

In this system, the vehicle itself becomes a stabilized platform. The gyros are hard mounted to the vehicle and provide signals which are used for direct torquing of the vehicle around the body axes. The gyros are used in a position mode except when reset of the reference is required. The rate mode is used for ease in reset. Control of vehicle rotations required for alignment and horizontal thrust control is supplied by the computer or can be a manual input. These rotations are controlled by application of a stabilized torquing current to the proper gyro. The torquer current defines a fixed gyro error signal. The error signal is processed to give a controlled vehicle rotation rate. A timed application of the torquer current, therefore, determines the angle of vehicle rotation.

The accelerometer monitors the thrust applied and each pulse of output represents a fixed increment of velocity. Control can be achieved either by counting pulses or by measuring pulse rate. The first measures velocity change directly, while the second provides a measure of acceleration. Both manual and computer measurements can be easily attained.

The computer-sequencer is the central agent for providing automatic vehicle control. During manual control it will act as a bookkeeper to keep track of time, distance to go, and present velocity, and it may also be used to provide warning when something is wrong.

The telescope and altimeter are peripheral devices used for initial alignment and pilot assist. It would be possible to use the telescope to control attitude during the flight phases but this would add complication, especially in the computer.

4.2 HARDWARE DESCRIPTION

The following is a summary of the estimated weights, power requirements and volume of the equipment required:

	Weight (Pounds)	Power (Watts)	Volume (cu. in)
Gyro and accelerometer package	10	21	240
Computer-sequencer	6	8	360
Displays and manual controls	5	1	800
Command processing	4	6	100
Telescope and mount	5	-	150
Altimeter (includes antenna)	8	26	340
Total	38	62	1990

The above weight and power figures are projections from existing hardware.

A gyro and accelerometer package presently exists which has specifications similar to those required for the LFV. This package, built by Nortronics for JPL, is now on its way to Mars on the Mariner spacecraft. It weighs 10.9 pounds and requires 21 watts. All necessary electronics are included. The computed mean-time to failure for the electronics plus the accelerometer is 21,000 hours. An upgrading of the electronics using integrated circuits and a different choice of gyros will bring the weight down and increase the MTBF.

The computer sequencer will use integrated circuits and a core memory. Units with similar requirements (also built by Nortronics for Ranger) for processing and sequencing presently exist. These units, using conventional printed circuits, weigh 9 pounds and this should be reduced at least one-third by the use of integrated circuits.

The Command Processing module contains the electronics required to convert commands into vehicle torquing signals. This would utilize integrated circuits and/or cordwood construction. This module may be combined with others for packaging convenience.

A proposed altimeter for the Apollo was used as a guide to estimation of weight and power required. This altimeter had a maximum range of 20 n. miles. The power required was scaled down by 50% since the present desired range is only 20,000 feet. This power estimate may still be high.

SECTION 5.0

ERROR ANALYSIS

A simplified error analysis is enclosed for both ballistic and hover-translation flights using off-the-shelf equipment accuracies. Accelerometer scale factor contributed a sizeable error, and better accelerometers are available. Initial alignment is a much more difficult problem to solve. The decreased lunar gravity makes both bubble levels and accelerometers less sensitive to horizontal alignment errors. The slow spin rate of the moon makes gyro-compassing unfeasible. Automatic star trackers are required to significantly reduce the aximuth alignment errors. A 2-sigma CEP of 0.25 miles is not possible unless the initial orientation errors can be reduced a great deal.

The gyros and accelerometer assumed for the analysis are those used in the Ranger project and are felt to be representative of equipment available for this type of service. The component accuracies are as follows:

K_0	=	.0075 ft/sec	accelerometer null bias
K_1	=	0.0025	accelerometer scale factor error
K_2	=	0.003	gyro scale factor error
K_{3i}	=	1°/hr	i gyro drift i = 1, 2, 3 corresponding to x, y, z
ϕ_x	=	0.1°	x axis misalignment
ϕ_y	=	0.1°	y axis alignment
ϕ_z	=	0.2°	z axis misalignment (azimuth error)

It may be argued that this error analysis is overly pessimistic. The component errors assumed represent equipment designed 4 years or more ago and no improvement has been predicted. Further the vertical alignment errors are greater than obtainable even with bubble levels currently available. This criticism is justified if one assumes earth or even deep space conditions. However, the conditions of actual use on the lunar surface are more rigorous than any similar equipment has been subjected to in the past. It is believed that pessimism is better than optimism in this area.

There is a further reason for the pessimism regarding the vertical alignment errors. The gravitational figure of the moon is not known to any precision. There is speculation that the center of gravity is not aligned closely with the geometrical center. This view is expressed by astronomers who should be knowledgeable in this area. No analysis has yet been made of the errors from this source but they could be quite serious if the displacement is sizable.

All of the analysis has assumed a flat lunar model. In a 50 mile flight, however, the lunar vertical (selenodetic) rotates 2.6°. This will require a modification in the mechanization involving torquing of the pitch gyro to account for the rotation. Additional error will result from this source. While the mechanization is thereby complicated, no additional equipment is required.

5.1 ERROR EQUATIONS

Define a coordinate system with the x axis in the direction of flight, (See Figure 3.1), z axis vertical, and y axis in the horizontal plane oriented to form an orthogonal coordinate system. Velocity errors are:

$$\Delta V_x = \phi_y \text{ At} \cos 0_c + K_2 \phi_c \text{ At} \cos 0_c + \frac{K_{32}At^2}{2} \cos \phi_c + K_1 t \sin \phi_c + K_1 \text{ At} \sin \phi_c \tag{2}$$

$$\Delta V_y = - \phi_x \text{ At} \cos \phi_c + \phi_z \text{ At} \sin \phi_c - \frac{K_{31}At^2}{2} \cos \phi_c + \frac{K_{33}At^2}{2} \sin \phi_c \tag{3}$$

$$\Delta V_z = - \phi_y \text{ At} \sin \phi_c - K_2 \phi_c \text{ At} \sin \phi_c + \frac{K_{32}At^2}{2} \sin \phi_c + K_1 t \cos \phi_c + K_1 At \cos \phi_c \tag{4}$$

Where ϕ_c is the commanded thrust angle measured from the vertical. The error arising from each term is considered to be uncorrelated to the remaining terms. Accordingly, the initial velocity errors entering into position errors must be attributed to the correct source to avoid indicating correlations where none exist.

The position errors become:

$$\Delta S_x = V_{xo}(\phi_y)t + V_{xo}(K_2)t + V_{xo}(K_{32})t + V_{xo}(K_o)t + V_{xo}(K_1)t$$

$$+ \frac{\phi_y At^2}{2} \cos \phi_c + \frac{K_2 \phi_c At^2}{2} \cos \phi_c + \frac{K_{32} At^3}{6} \cos \phi_c$$

$$+ \frac{K_1 At^2}{2} \sin \phi_c \tag{5}$$

$$\Delta S_y = V_{yo}(\phi_x)t + V_{yo}(\phi_z)t + V_{yo}(K_{33})t$$

$$- \frac{\phi_x At^2}{2} \cos \phi_c + \frac{\phi_z At^2}{2} \sin \phi_c - \frac{K_{31} At^3}{6} \cos \phi_c$$

$$+ \frac{K_{33} At^3}{6} \sin \phi_c \tag{6}$$

$$\Delta S_z = V_{zo}(\phi_y)t + V_{zo}(K_2)t + V_{zo}(K_{32})t + V_{zo}(K_o)t + V_{zo}(K_1)t$$

$$- \frac{\phi_y At^2}{2} \sin \phi_c - \frac{K_z \phi_c At^2}{2} \sin \phi_c - \frac{K_{32} At^3}{6} \sin \phi_c$$

$$+ \frac{K_o t^2}{2} \cos \phi_c + \frac{K_1 At^2}{2} \cos \phi_c \tag{7}$$

where $V_{xo}(k)$ is the initial velocity error from scurce K.

5.2 BALLISTIC ERROR ANALYSIS

The miss sensitivities in parabolic flight are:

$$\frac{\partial S_x}{\partial V_x} = \frac{2V_{zi}}{gm} \tag{8}$$

$$\frac{\partial S_x}{\partial V_z} = \frac{2V_{xi}}{gm} \tag{9}$$

$$\frac{\partial S_y}{\partial V_y} = \frac{2V_{zi}}{gm} \tag{10}$$

Thus the miss distance at the end of parabolic flight in a ballistic trajectory is a function of the injection errors. The total error is the sum of the parabolic errors and the ascent, thrust, retro-thrust, and descent errors.

Equations (2) through (7) must be evaluated for each flight phase. For ballistic flight, the following values are assumed:

t_a	= 15	sec	ascent time
t_t	= 118	sec	thrust time
t_c	= 207	sec	coast time
t_r	= 103	sec	retro-fire time
t_d	= 15	sec	descent time
A_a	= 10.7	ft/sec^2	ascent acceleration
A_t	= 11.2	ft/sec^2	thrust acceleration
A_r	= 13.4	ft/sec^2	retro acceleration

A_d = 10.7 ft/sec^2 descent acceleration

ϕ_c = 37° thrust angle

V_{zi} = 550.8 ft/sec vertical injection velocity

V_{xi} = 828.0 ft/sec horizontal injection velocity

Table 5.1 gives the results of the ballistic error analysis. The velocity entries in the coast phase are not errors generated during this phase, but rather the errors accumulated up to the beginning of the phase. In the remaining phases, the velocity errors shown are those generated in that phase. An initial position error was not assumed. If a 100 foot error exists, the CEP will increase to 1872 feet.

5.3 HOVER-TRANSLATION ERROR ANALYSIS

A hover-translation flight at 1300 feet and 900 fps was assumed for an error analysis. The times and average accelerations are as follows:

t_a = 15 sec ascent time

t_t = 88.25 sec thrust time

t_c = 168 sec coast time

t_r = 160 sec retro time

t_d = 13.7 sec descent time

A_a = 10.7 ft/sec^2 ascent acceleration

A_t = 11.4 ft/sec^2 retro acceleration

A_c = 5.32 ft/sec^2 coast acceleration

A_r = 7.74 ft/sec^2 retro acceleration

A_d = 8.55 ft/sec^2 descent acceleration

TABLE 5.1
BALLISTIC ERROR ANALYSIS

	ΔS_x ft	ΔS_y ft	ΔV_x fps	ΔV_y fps	ΔV_z fps
Ascent $\phi c=0°$					
Null bias	-	-	-	-	.048
Acc scale	-	-	-	-	.803
Gyro scale	-	-	-	-	-
x gyro drift	-	0.0	-	.005	-
y gyro drift	0.0	-	.005	-	-
z gyro drift	-	-	-	-	-
x tilt	-	-2.1	-	.280	-
y tilt	2.1	-	.280	-	-
Azimuth error	-	-	-	-	-
Thrust $\phi c=37°$					
Null bias	31.5	-	.533	-	.707
Acc scale	117.2	-	1.986	-	2.635
Gyro scale	120.6	-	2.044	-	-1.540
x gyro drift	-	-10.4	-	-.251	-
y gyro drift	10.4	-	.251	-	-.189
z gyro drift	-	7.4	-	.189	-
x tilt	-	-141.7	-	-1.842	-
y tilt	141.7	-	1.842	-	-1.388
Azimuth error	-	163.8	-	2.776	-
Coast					
Null bias	345.4	-	.533	-	.755
Acc scale	858.8	-	1.986	-	1.438
Gyro scale	-56.2	-	2.044	-	-1.540
x gyro drift	-	-53.0	-	.256	-
y gyro drift	-5.8	-	.256	-	-.189
z gyro drift	-	39.1	-	.189	-

TABLE 5.1 (Continued)

	ΔS_x ft	ΔS_y ft	ΔV_x fps	ΔV_y fps	ΔV_z fps
Coast (con't)					
x tilt	-	-439.4	-	-2.122	-
y tilt	7.3	-	2.122	-	-
Azimuth error	-	574.8	-	2.776	-
Retro					
Null bias	31.0	-	-.465	-	.617
Acc scale	97.8	-	-2.073	-	2.751
Gyro scale	320.4	-	2.134	-	1.608
x gyro drift	-	-18.5	-	.229	-
y gyro drift	34.3	-	.229	-	.172
z gyro drift	-	13.6	-	-.172	-
x tilt	-	-119.6	-	1.923	-
y tilt	317.6	-	1.923	-	1.449
Azimuth error	-	136.7	-	-2.898	-
Descent					
Null bias	1.0	-	-	-	.048
Acc scale	-1.3	-	-	-	.803
Gyro scale	62.7	-	-	-	-
x gyro drift	-	0.4	-	-.005	-
y gyro drift	7.3	-	.005	-	-
z gyro drift	-	0.3	-	-	-
x tilt	-	-5.1	-	-.280	-
y tilt	62.8	-	.280	-	-
Azimuth	-	1.8	-	-	-

TABLE 5.1 (Continued)

	ΔS_x ft	ΔS_y ft	ΔV_x fps	ΔV_y fps	ΔV_z fps
Total					
Null bias	408.9	-	.068	-	1.420
Acc scale	1072.5	-	-.087	-	4.992
Gyro scale	447.5	-	4.178	-	.068
x gyro drift	-	-81.5	-	-.490	-
y gyro drift	46.3	-	.490	-	-.017
z gyro drift	-	60.4	-	.017	-
x tilt	-	-705.8	-	-.479	-
y tilt	529.4	-	4.325	-	.064
Azimuth	-	877.1	-	-.122	-
RSS Total	1342	1130			
CEP	1450				

Table 5.2 contains the summary of the hover-translation errors. Vertical errors were not considered since they do not contribute to miss distance in this case. Again note that 1000 feet initial position error in each axis will increase the CEP to 2025 feet. It is evident that the hover-translation mode is slightly less accurate than the ballistic mode when the same component accuracies are assumed. The initial alignment errors and the accelerometer scale factor errors are the largest miss contributors in both cases.

Figure 5.1 shows the probability of falling outside a circle of given radius, based on the probability ellipses resulting from the errors shown above.

TABLE 5.2

HOVER TRANSLATION ERROR ANALYSIS

	ΔS_x ft	ΔS_y ft	ΔV_x fps	ΔV_y fps
<u>Ascent</u>				
Null bias	-	-	-	-
Acc scale	-	-	-	-
Gyro scale	-	-	-	-
x gyro drift	-	0.0	-	-.005
y gyro drift	0.0	-	.005	-
z gyro drift	-	-	-	-
x tilt	-	-2.1	-	-.280
y tilt	2.1	-	.280	-
Azimuth error	-	-	-	-
<u>Thrust</u>				
Null bias	25.6	-	.580	-
Acc scale	97.2	-	2.204	-
Gyro scale	68.5	-	1.553	-
x gyro drift	-	-.7	-	-.010
y gyro drift	.7	-	.010	-
z gyro drift	-	.5	-	.018
x tilt	-	-62.0	-	-.846
y tilt	62.0	-	.846	-
Azimuth error	-	135.8	-	3.077
<u>Coast</u>				
Null bias	97.4	-	-	-
Acc scale	370.3	-	-	-
Gyro scale	260.9	-	-	-
x gyro drift	-	-22.9	-	-.364
y gyro drift	22.9	-	.364	-
z gyro drift	-	3.0	-	-

TABLE 5.2 (Continued)

	ΔS_x ft	ΔS_y ft	ΔV_x fps	ΔV_y fps
Coast (continued)				
x tilt	-	3.0	-	-1.560
y tilt	320.1	-	1.560	-
Azimuth error	-	516.9	-	-
Retro				
Null bias	25.3	-	-.844	-
Acc scale	178.4	-	-2.178	-
Gyro scale	413.3	-	2.060	-
x gyro drift	-	-78.5	-	-.338
y gyro drift	78.5	-	.338	-
z gyro drift	-	-14.8	-	-.334
x tilt	-	-552.7	-	-1.536
y tilt	552.7	-	1.536	-
Azimuth error	-	249.1	-	-3.040
Descent				
Null bias	-3.7	-	-	-
Acc scale	.4	-	-	-
Gyro scale	49.5	-	-	-
x gyro drift	-	-9.9	-	0.0
y gyro drift	9.9	-	0.0	-
z gyro drift	-	-4.4	-	-
x tilt	-	-59.3	-	-.204
y tilt	59.3	-	.204	-
Azimuth error	-	.5	-	-
Total				
Null bias	144.6	-	-	-
Acc scale	646.3	-	-	-
Gyro scale	792.7	-	-	-

TABLE 5.2 (Continued)

	ΔS_x ft	ΔS_y ft	ΔV_x fps	ΔV_y fps
<u>Total</u> (continued)				
x gyro drift	-	-112.0	-	-
y gyro drift	112.0	-	-	-
z gyro drift	-	-15.7	-	-
x tilt	-	-996.2	-	-
y tilt	996.2	-	-	-
Azimuth error	-	902.3	-	-
RSS Total	1439	1349		
CEP	1645			

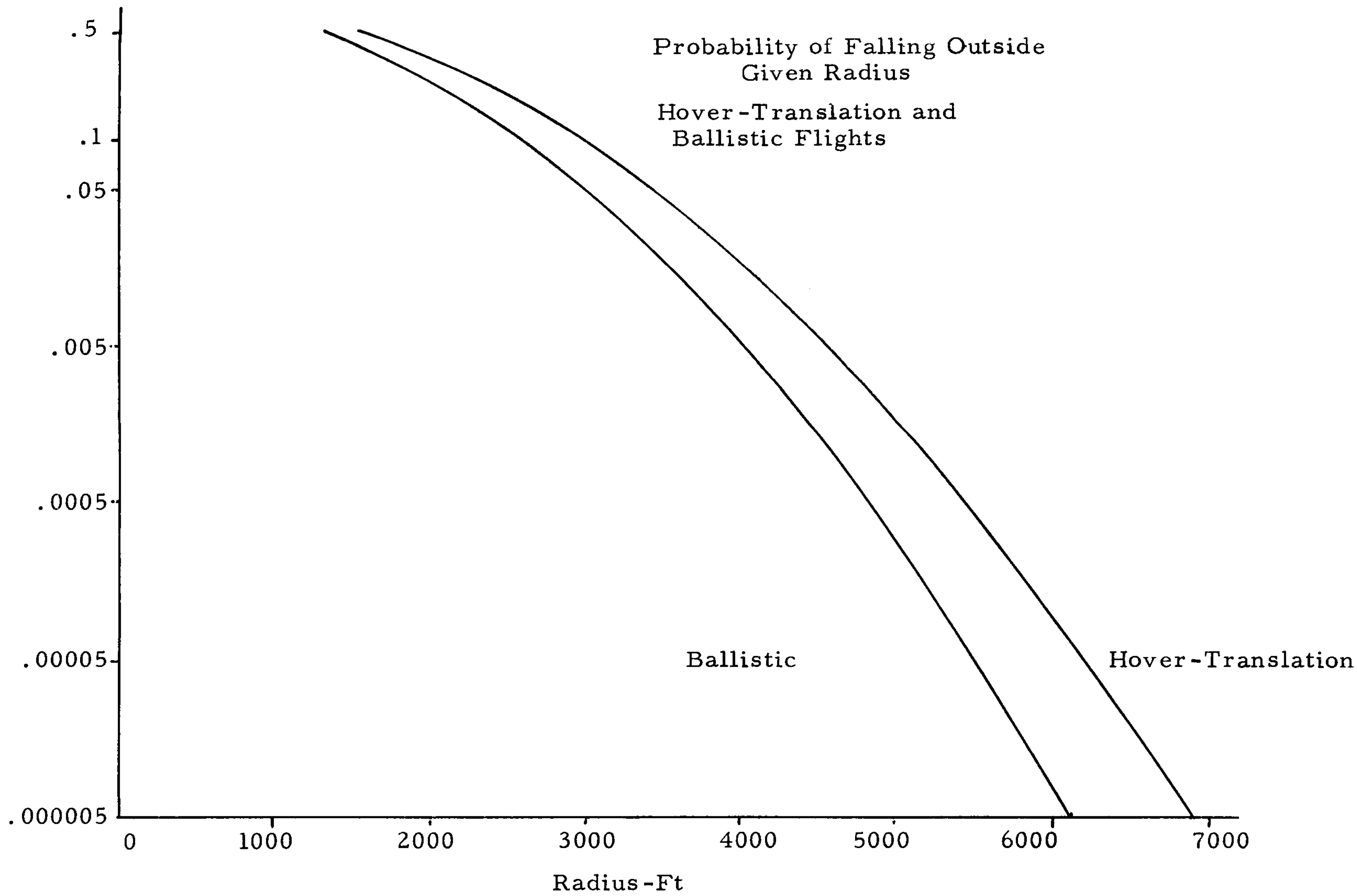

Probability of Falling Outside
Given Radius
Hover-Translation and
Ballistic Flights
.5
.1
.05
.005
.0005
.00005
.000005
0
1000
2000
3000
4000
5000
6000
7000
Radius-Ft
Ballistic
Hover-Translation

Figure 5.1

SECTION 6.0

CONCLUSIONS AND RECOMMENDATIONS

1. Using the vehicle as defined by the guidelines, a simple guidance and navigation system can perform the required 50 mile mission within the 4700 fps. V allowance using a transit time of less than 15 minutes. There will be sufficient fuel to provide terminal error corrections, even assuming the system errors exceed the 0.25 mile error specified. Additional terminal aids other than the optical aids permitted by the guidelines are desirable and probably necessary.

2. Full manual operation is practical using a minimum of equipment. Further study is required to determine if the manual mode is desirable during the early flights.

3. A semi-automatic mode of operation is advisable for astronaut rescue and requires minor additions to the guidance and navigation system.

4. The desired accuracy of 0.25 miles cannot be achieved with a simple mechanization assuming realistic equipment errors.

5. Terminal aids are required. For daytime use an optical aid may not be sufficient.

6. The hover-translation trajectory is more efficient in fuel usage but is slightly less accurate than the ballistic trajectory.

7. Modifications to the hover-translation trajectory provide greater fuel efficiency but may complicate the mechanization beyond the point of acceptance.

8. Use of the ballistic trajectory should be ruled out due to the possible failure of the engines to re-ignite for retro. This is considered to be the most likely failure mode for the engine.

9. Further study is required to determine trajectory modifications which provide more optimum fuel usage. A more sophisticated computer program will be required to complete this study.

10. Coupled with the preceding recommendation, mechanization studies are required to select those trajectories which result in simple mechanizations.

11. Further study of the fuel requirements is required in order to choose between the constant acceleration (variable fuel flow-rate) boost profile and the constant thrust (variable pitch-over) profile. The first profile gives the simplest mechanization.

12. A dynamical study of the guidance system, the control system and the vehicle is required. This study would provide a guide for the development of specifications which properly reflect the dynamical interplay between these three systems. Only by this means can efficient mechanization for each system be attained.

SECTION 7.0

REFERENCES

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SECTION 8.0

DEFINITION OF SYMBOLS

a - thrust acceleration

g - lunar gravity

g_e - earth gravity

g_m - lunar gravity

k_o - accelerometer bias error

k_1 - accelerometer scale factor error

k_2 - gyro scale factor error

k_{3i} - i gyro drift rate, i = x, y, z

m_o - initial vehicle weight

m_{oc} - initial vehicle weight, coast phase

m_{od} - initial vehicle weight, descent phase

m_{or} - initial vehicle weight, retro phase

m_{ot} - initial vehicle weight, thrust phase

m_f - total fuel consumed

m_{fa} - fuel consumed, ascent phase

m_{fc} - fuel consumed, coast phase

m_{fd} - fuel consumed, descent phase

m_{fr} - fuel consumed, retro phase

m_{ft} - fuel consumed, thrust phase

$\dot{m}$ - fuel mass flow rate

$\dot{m}_d$ - mass flow rate, descent

$\dot{m}_r$ - mass flow rate, retro

t - time

ta - ascent time

tc - coast time

td - descent time

th - hover or coast time

tr - retro time

t_t = thrust time

u - dummy probability variable along major axis of error ellipse

v - dummy probability variable along minor axis of error ellipse

v - translation velocity

x - down-range miss

y - cross-range miss

H - altitude

Isp - specific impulse

K - probability parameter corresponding to number of standard deviations

L - loss parameter

La - ascent loss parameter

Ld - descent loss parameter

M_o - initial weight

M_f - final weight

P - probability

P_R - probability of acquisition

P_S - probability of success

Q - fuel consumed

R_K - radius of visibility, or RF line-of-sight

S_o - distance to be travelled by hover-translation

S_o - initial position

S_x - x position component (down-range)

S_{xc} - down-range distance during coast phase

S_{xr} - down-range distance during retro phase

S_{xt} - down-range distance during thrust phase

S_y - y position component (cross-range)

S_z - z position component (altitude)

S_{zc} - altitude gained by coasting upward

S_{zt} - altitude gained by thrusting

V - velocity

Vo - initial velocity

Ve - effective exhaust velocity

Vopt - optimum translation velocity

Vs - vertical velocity neglecting gravity loss

Vx - x velocity component

Vx_o - initial x velocity component

Vy - y velocity component

Vz - z velocity component

Vz_o - initial z velocity component

θ - polar varialbe of integration of probability ellipse

ρ - correlation coefficient between down-range and cross-range miss

σ_1 - standard deviation along major axis of error ellipse

σ_2 - standard deviation along minor axis of error ellipse

σ_x - down-range standard deviation

σ_y - cross-range standard deviation

ϕ - rotation angle to make coordinates correspond to principal axis of error ellipse

ϕ - thrust angle measured from the vertical

ϕ_c - commanded thrust angle

ϕ_x - misalignment about x axis

ϕ_y - misalignment about y axis

ϕ_z - misalignment about z axis (azimuth error)

ΔSx - down-range position error

ΔSy - cross-range position error

ΔSz - vertical position error

ΔVx - downrange velocity error

ΔVy - cross-range velocity error

ΔVz - vertical velocity error

APPENDIX A

DERIVATION OF TRAJECTORY EQUATIONS

The acceleration, or thrust, of a rocket engine is:

$$a = \frac{\dot{m} g_o \; Isp}{m_o - \dot{m}t} \tag{1}$$

where
a = thrust
$\dot{m}$ = fuel mass flow rate
go = earth gravity
Isp = specific thrust
m_o = initial vehicle weight
t = time

Some saving in symbols can be made if exhaust velocity is defined.

$$Ve = g_o \; Isp \tag{2}$$

Velocity is given by the integral of (1).

$$V = Vo + Ve \ln \left(\frac{m_o}{m_o - \dot{m}t} \right) \tag{3}$$

A second integration yields distance.

$$S = So + Vot + Ve \left[\left(\frac{\dot{m}t - m_o}{\dot{m}} \right) \ln \left(\frac{m_o}{m_o - \dot{m}t} \right) + t \right] \tag{4}$$

Introducing a coordinate system with x horizontal and z vertical, the components of velocity and distance with the gravity loss and fixed thrust angle are:

$$Vx = Vx_o + Ve \sin \emptyset \ln \left(\frac{m_o}{m_o - \dot{m}t} \right) \tag{5}$$

$$V_z = Vz_o + Ve \cos \emptyset \ln \left(\frac{m_o}{m_o - \dot{m}t} \right) - g_m t \tag{6}$$

$$S_x = S_{xo} + V_{xo}t + Ve \sin \emptyset \left[\left(\frac{\dot{m}t - m_o}{\dot{m}} \right) \ln \left(\frac{m_o}{m_o - \dot{m}t} \right) + t \right] \tag{7}$$

$$S_z = S_{zo} + V_{zo}t + V_e \sin \emptyset \left[\left(\frac{\dot{m}t - m_o}{\dot{m}} \right) \ln \left(\frac{m_o}{m_o - \dot{m}t} \right) + t \right] - \frac{g_m t^2}{2} \tag{8}$$

Where $\emptyset$ = angle of thrust from vertical

g_m = moon gravity

For vertical rise, the velocity equation is separated into thrust and gravity components

$$Vs = Ve \ln \left(\frac{m_o}{m_o - \dot{m} t} \right)$$

$$Vz = Vs - g_m t_a \qquad (9)$$

where t_a is the thrust time for ascent.

The vertical distance becomes:

$$Sz_t = Vs \left(\frac{\dot{m} t - m_o}{\dot{m}} \right) + Ve\ t_a - \frac{g_m t_a^2}{2} \qquad (10)$$

The altitude gained by coasting until the velocity is zero is

$$Sz_c = \frac{V_z^2}{2 g_m} \qquad (11)$$

$$Sz = Sz_t + Sz_c$$

Since a constant flow rate is assumed, the fuel used is:

$$m_{fa} = \dot{m} t \qquad (12)$$

The angle from the vertical that the thrust vector can be pointed to give a vertical component equal to gravity is:

$$\cos \emptyset = \frac{g_m m_{ot}}{\dot{m}\ Ve} \qquad (13)$$

The horizontal velocity equation can be solved for the time necessary to achieve a velocity, assuming constant flow rate and thrust angle.

$$t_t = \frac{m_{ot}}{\dot{m}} \left[1 - \exp \left(\frac{-Vx}{Ve \sin \emptyset} \right) \right] \tag{14}$$

The distance travelled is

$$Sx_t = \left(\frac{\dot{m}t - m_{ot}}{\dot{m}} \right) V_x + Vet \sin \emptyset \tag{15}$$

The fuel consumed is

$$m_{ft} = \dot{m}\ t_t \tag{16}$$

The flow rate is variable during coast, and must provide sufficient acceleration to null gravity

$$\dot{m} = \frac{g_m m_{oc}}{Ve + g_m t_c} \tag{17}$$

where m_{oc} is the vehicle weight at the start of coast.

If coast is assumed to last for t_c seconds, the fuel consumed will be:

$$m_{fc} = m_{oc} \left[1 - \exp \left(\frac{-g_m t_c}{Ve} \right) \right] \tag{18}$$

The distance travelled will be

$$Sx_c = Vx\ t_c \tag{19}$$

A new flow rate, $\dot{m}_r$, is assumed for retro-fire.

The initial angle, $\emptyset$, is assumed constant to simplify the program and is:

$$\cos \emptyset = \frac{g_m m_{or}}{\dot{m}_r Ve} \tag{20}$$

The time required to null the horizontal velocity is:

$$t_r = \frac{m_{or}}{\dot{m}_r} \left[1 - \exp\left(\frac{-Vx}{Ve \sin \emptyset}\right) \right] \tag{21}$$

and the distance travelled is:

$$S_{xr} = Vx\, t_r - \left(\frac{\dot{m}_r t_r - m_o}{\dot{m}_r}\right) Vx - Ve\, t_r \sin \emptyset \tag{22}$$

A flow rate, $\dot{m}_d$, is assumed for descent. The time that this flow rate must exist to null the velocity gained in falling from height S_z is

$$t_d = \frac{m_{od}}{\dot{m}_d} \left[1 - \exp\left(\frac{-\sqrt{2g_m S_z}}{Ve}\right) \right] \tag{23}$$

and the fuel consumed is

$$m_{fd} - \dot{m}_d t_d \tag{24}$$

The fuel used and distance travelled is totalled:

$$m_f = m_{fa} + m_{ft} + m_{fc} + m_{fr} + m_{fd} \tag{25}$$

$$Sx = S_{xt} + S_{xe} + S_{xr} \tag{26}$$

Trial flights are made with varying coast times until the desired distance is covered. This program is by no means optimum, but gives a quick means of finding the upper bound on the fuel required based on the altitude, flow rates, and distance covered.

The values used for the simulation were:

m_o = 1600 lb. initial LFV weight

g_o = 32.16 fps^2 earth gravity

g_m = 5.32 fps^2 lunar gravity

Isp = 300 sec specific thrust

$\dot{m}$ = 1.8 lb/sec maximum fuel flow rate

$\dot{m}$ = 0.6 lb/sec minimum fuel flow rate

A flat moon with a uniform gravitational field was assumed. An altimeter would provide altitude and allow the astronaut to manually control altitude changes.

APPENDIX B

DERIVATION OF OPTIMUM VELOCITY FOR HOVER-TRANSLATION

For rectilinear flight in gravity free space the velocity achieved starting from rest is:

$$\overline{V} = g_e \, Isp \log \frac{M_o}{M_f} \qquad (1)$$

where g_e = 32.2 ft/sec^2

Isp = specific impulse

M_o = initial mass

M_f = final mass

$(M_o = M_f)$ is the mass of fuel burned.

Clearly $\overline{V}$ increases monotonically with the amount of fuel burned.

For non-rectilinear flight in a gravity field the achieved velocity V does not equal the free space velocity $\overline{V}$. A gravity and turning loss parameter L can be defined such that

$$V = (1-L)\overline{V} \qquad (2)$$

Strictly speaking L is a function of the burnout altitude and velocity as well as the path followed. If the minimum fuel path is followed then L depends only on the burnout altitude and velocity. For the lunar flying vehicle it is possible to define an ascent loss parameter L_a and a descent loss parameter L_d.

Simple Hover Translation - For the lunar flying vehicle a type of trajectory between two points can be described simply as follows: The vehicle is delivered to an altitude H with a horizontal velocity V such that it continues by hover translation. At the end of the hover translation the vehicle is delivered to altitude zero with zero velocity. The complete trajectory, then, is characterized by three sections: ascent, hover translation, and descent. The minimum fuel pitch program for the ascent and descent sections is given by the linear tangent law tan $\phi = a + bt$. The translational velocity V must be chosen to give maximum range for the fuel used or equivalently to require minimum fuel for the intended range.

A little consideration shows that the total equivalent free space velocity potential required for all three sections is:

$$\bar{V}_{tot} = \frac{V}{1-L_a} + g t_h + \frac{V}{1-L_d} \qquad (3)$$

where g = lunar gravity acceleration

t_h = hover time

The hover distance $S_o = V t_h$ thus

$$\bar{V}_{tot} = \frac{V}{1-L_a} + \frac{g S_o}{V} + \frac{V}{1-L_d} \qquad (4)$$

Using (1) it is clear that the total fuel Q consumed is just:

$$Q = M_o - M_f = M_o \left\{ 1 - \exp\left(- \frac{\bar{V}_{tot}}{g_e I_{sp}} \right) \right\} \qquad (5)$$

Given S_o it is desired to choose V such that Q is minimized; however since Q grows monotonically with $\bar{V}_{tot}$ it is clear that the problem amounts to minimizing $\bar{V}_{tot}$. Taking the derivative of (4) yields the optimum V:

thus

$$0 = \frac{d\bar{V}_{tot}}{dv} = \frac{1}{1-L_a} - \frac{g S_o}{V^2} + \frac{1}{1-L_d}$$

$$V_{opt} = \sqrt{\frac{g S_o (1-L_a)\ (1-L_d)}{2-L_a-L_d}} \qquad (6)$$

(In taking the derivative it has been assumed that L_a and L_d do not depend on V. It is expected that the error incurred is negligible.)

APPENDIX C

DERIVATION OF TARGET ERROR PROBABILITIES

Letting σ_x and σ_y be the downrange and crossrange standard deviations, and ρ the correlation coefficient between downrange and crossrange miss the probability of (x, y) being within a circle of radius R_k is

$$P\left(\sqrt{x^2+y^2} \leqslant R_k\right) = P(x^2+y^2 \leqslant R_k) = \iint P(x,y)\,dx\,dy \qquad (1)$$

where P(x, y) is the bivariate normal distribution

$$P(x,y) = \frac{1}{2\pi\sigma_x\sigma_y\sqrt{1-\rho^2}} \exp\left\{-\frac{1}{2(1-\rho^2)}\left(\frac{x^2}{\sigma_x^2} - \frac{2\rho xy}{\sigma_x\sigma_y} + \frac{y^2}{\sigma_y^2}\right)\right\} \qquad (2)$$

If the probability of visual acquisition is uniformly distributed with probability

$$P_R = \begin{cases} 0 & x^2+y^2 > R_k \\ P_R & x^2+y^2 < R_k \end{cases} \qquad (3)$$

The probability of success is then

$$P_S = P_R\, P(x^2+y^2 \leqslant R_k) \qquad (4)$$

This under the assumption that the LFV has sufficient maneuvering capability to reach the target once visual acquisition of the target is achieved.

The integral to be performed in equations (1) and (2) can be simplified by transforming to (u, v) variables which is a rotation to the principal axis through the angle $\emptyset$. Then

$$P(u,v) = \frac{1}{2\pi\sigma_1 \sigma_2} \exp\left\{-\frac{1}{2}\left(\frac{u^2}{\sigma_1^2} + \frac{v^2}{\sigma_2^2}\right)\right\} \tag{5}$$

where

$$\sigma_1^2 = \frac{(1-\rho^2)\,\sigma_x^2\,\sigma_y^2}{\sigma_y^2\cos^2\emptyset + \sigma_x^2\sin^2\emptyset - \rho\sigma_x\,\sigma_y\,\sin 2\,\emptyset}$$

$$\sigma_2^2 = \frac{(1-\rho^2)\,\sigma_x^2\,\sigma_y^2}{\sigma_x^2\cos^2\emptyset + \sigma_y^2\sin^2\emptyset + \rho\sigma_x\,\sigma_y\,\sin 2\,\emptyset} \tag{6}$$

Normalizing and transforming to polar coordinates we have

$$\frac{u}{\sigma_1} = K\cos\theta$$

$$\frac{v}{\sigma_2} = K\sin\theta \tag{7}$$

$$P(K,\theta) = \frac{K}{2\pi}\,e^{-\frac{1}{2}K2}$$

Hence

$$P_S = \frac{P_R}{2}\int_0^{2\pi}\int_0^{\frac{R_K}{\sigma_1^2\cos^2\theta + \sigma_2^2\sin^2\theta}} Ke^{-\frac{K2}{2}}\,dK\,d\theta \tag{8}$$

or

$$P_S = P_R\left(1 - \frac{1}{\pi}\int_0^{\pi} \exp\left\{\frac{-R_K^{\ 2}}{\sigma_1^{\ 2} + \sigma_2^{\ 2} - (\sigma_2^{\ 2} - \sigma_1^{\ 2}) \cos^2 \theta}\right\} d\theta\right) \tag{9}$$

An IBM 7090 program has been developed for purposes of evaluating equation (9). This digital program will be used for evaluating P_S for future LFV study work.

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